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A Note on the Estimation of Some Low Speed Characteristics of Delta Wings*

-by-

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SUMMARY

The results of the Weissinger swept lifting line theory have been compared with other methods and experimental results. On the basis of this comparison empirical corrections to the Weissinger theory results have in some cases been suggested. Charts have been prepared which enable an estimate to be made of the lift curve slope at zero lift, the location of the aerodynamic centre, and the rolling and yawing characteristics at low incidence. The estimation of C_{Lmax} and $\alpha_{C_{Lmax}}$ are also briefly considered.

Great accuracy cannot be claimed for the resulting diagrams, but it is considered that they provide a rapid method of estimating the above mentioned characteristics of delta wings, both cropped and uncropped, to an accuracy that is adequate in the design stage of an aeroplane.

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NOTATION

$\Lambda_{\frac{1}{4}}$	Sweepback of 1/4 chord line, degrees.
A	Aspect ratio ($= b^2/S$).
c_r	Root chord, ft.
c_t	Tip chord, ft.
S	Gross wing area, sq.ft.
λ	Taper ratio ($= c_t/c_r$).
s	Semi-span ($= b/2$), ft.
x	Longitudinal coordinate, ft.
y	Spanwise coordinate, ft.
η	Non-dimensional spanwise coordinate ($= y/s$).
c	Local chord, ft.
$\bar{c}$	Standard (or geometric) mean chord, ft.
$\bar{\bar{c}}$	Centroid-of-area chord, ft.
C_L	Total lift coefficient for whole wing.
$x_{A.C.} \cdot \bar{\bar{c}}$	Distance of A.C. aft of the L.E. of C.A.C., ft.
ℓ_p	Damping in roll derivative.
a_o	Average section lift curve slope, per rad.
n_{r_i}	Yawing moment due to yawing derivative (induced drag component).
n_p	Yawing moment due to rolling derivative.

1. Introduction

In this note the description 'delta wing' includes the cropped delta wing, i.e. it is taken to apply to sweptback, straight tapered wings of triangular, or modified triangular planform with the apex forward, and with the trailing edge normal to the centre line of the wing. For delta wings as defined above a relationship between the planform parameters (viz. sweep, taper and aspect ratio) can be deduced as follows.-

$$\tan \Lambda_{\frac{1}{4}} = \frac{3(1-\lambda)}{A(1+\lambda)} \qquad \dots\dots\dots\dots (1)$$

This expression shows that a variation in any one of the three planform parameters has an effect on the other two. Thus, in assessing the characteristics of such wings, it is impossible to separate the effects of these parameters.

Existing lifting surface methods[1,2,3,4] of obtaining the characteristics of delta wings are both lengthy and laborious, and, in consequence, little systematic information has been published regarding the calculated changes in the characteristics with alteration in the planform parameters. The object of this investigation was to analyse the existing data, both theoretical and experimental, to see whether they lent themselves to the production of data sheets of an acceptable accuracy in the design stage of an aeroplane.

The method adopted was to start with the results of the Weissinger swept lifting line method[6,7] and to compare them with the results of other methods and with experimental results. In some cases this comparison has indicated a plausible empirical correction to the Weissinger theory results, and this correction has accordingly been adopted. Clearly, there are limitations in the results of such an analysis but, in general, it is believed that they are of satisfactory accuracy.

The characteristics that have been considered are the lift curve slope at zero lift, the location of the aerodynamic centre, and the rolling and yawing characteristics at low incidence. The estimation of $C_{L_{max}}$ and $\alpha_{C_{L_{max}}}$ has also been briefly considered.

2. Lift Curve Slope at $C_L = 0$

De Young[7] has calculated the lift curve slope for a wide range of sweptforward and sweptback straight tapered wings using the Weissinger method. Together with the relationship given in equation 1 use has been made of these results to prepare a 'carpet' in which lift curve slope is plotted against sweepback angle for a range of taper and aspect ratios. This carpet is shown in figure 1.

A comparison[8] of various methods of estimation includes but a few figures for the delta configuration, and to substantiate to curves of fig. 1, values given by them have been compared with as many theoretical and experimental results as were available. Lift curve slopes as predicted by the Weissinger method and by lifting surface theories are tabulated in table 1 and are plotted in figure 2 in the form of an accuracy curve. In the majority of cases excellent agreement is seen. When compared with experimental values (table 2 and figure 2) the Weissinger theory gives lift curve slopes agreeing to within 5 per cent for wings of aspect ratio greater than 1.5. For smaller aspect ratios the disagreement is somewhat greater, but the values can still be predicted to within about 8 or 9 per cent. Some scatter among the experimental results is to be expected since they were drawn from several reports describing tests made under a variety of conditions and at Reynolds numbers varying from 0.3×10^6 to 4.1×10^6. Little information is available concerning scale effect on the lift curve slope of delta wings, but, from tests[17] on wings 10 per cent thick, it would appear that an increase of 2 to 5 per cent is to be expected over the Reynolds number range from 0.5×10^6 to 1×10^7.

3. Characteristics at the Stall

In this section an analysis has been made of wind tunnel data to illustrate the effect of changes in planform. The data have been drawn from a variety of sources and some scatter of the experimental results is inevitable. Further the data were by no means comprehensive, so that the deductions can only be regarded as tentative. However, it is felt that the trends suggested are broadly speaking correct and significant.

3.1. Maximum Lift Coefficient and Stalling Incidence

From the results given in table 3 the variation of $C_{L\max}$ and $\alpha_{C_{L\max}}$ with sweepback for triangular wings of zero taper are plotted in figures 3 and 4.

For wings of moderate sweepback and fairly large aspect ratio (3.6 say) it would appear that $C_{L\max} \doteqdot 0.9$ can be expected. But with increase in sweepback to about 60° and the consequent reduction in aspect ratio the value of $C_{L\max}$ is increased to 1.2 to 1.3. This occurs when the angle of sweep is about 60°. At very large sweepback angles and very small aspect ratios $C_{L\max}$ is again of the order of 0.9. Similarly, the stalling incidence is a maximum when the sweepback angle is about 65°.

3.2. Lift Curve Slope up to the Stall

Some typical lift curves for delta wings are given in figure 5. The increasing non-linearity with reduction in aspect ratio is well illustrated. Comparison of the three families of curves indicate that the non-linearity of the lift curve is relatively little affected by changes of taper and sweep. A similar trend can be noticed in the results of some tests[20] of rectangular wings of small aspect ratio.

3.3. Longitudinal Stability near the Stall

The results of table 3 have been plotted in figure 6 after the manner of Shortal and Maggin.[21] Thus, figure 6 is a chart from which the longitudinal stability characteristics near the stall can be predicted. Owing to the limited extent of the available data concerning heavily swept delta wings of low aspect ratio the suggested margin between stable and unstable configurations near the stall can only be regarded as tentative.

4. Location of Aerodynamic Centre

In this analysis of results concerning the location of the aerodynamic centre of a delta wing, the leading edge of the centroid of area chord has been chosen as the reference point and distances are quoted as fractions of this chord length. For the sake of completeness, definitions of this chord, the standard mean chord and relations between the two are given in an appendix to the report.

The results of reference 7 have been used to prepare the carpet shown in figure 7 from which a first estimate of the position of the aerodynamic centre for any delta wing can be made. The values of $x'_{A.C.}$ given in figure 7 have been

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compared with those of as many experimental results and other theories as possible. The comparisons are given in figure 9 and table 4, and it is seen that in nearly every case the value of $x_{A.C.}$ has been underestimated. This is to be expected since an assumption in the derivation of figure 7 is that the locus of local aerodynamic centres follows the quarter chord line of the wing. In practice it has been shown[9,19] that at the root the local aerodynamic centre is well aft of the quarter chord point and at the tip it is slightly forward.

An analysis of these errors led to the construction of the set of curves given in figure 8. Thus, figures 7 and 8 can be used to estimate $x_{A.C.}$. An error curve for this modified estimate is shown in figure 10, and it is seen that, in general, the position of the aerodynamic centre can be predicted using figures 7 and 8 with a probable error of $\pm .01\ \bar{\bar{c}}$.

As an example of the method it is required to find the position of the aerodynamic centre of the following wing.-

$$A = 3.0\ ; \qquad \lambda = 0.05\ ; \qquad \Lambda_{\frac{1}{4}} = 42.2^{\circ}$$

Figure 7 shows $x'_{A.C.} = 0.318$ and

figure 8 gives $\Delta\, x'_{A.C.} \doteqdot 3.3$ per cent

$$\therefore \quad \Delta x'_{A.C.} = \frac{3.3}{100} \times .318 \doteqdot .011$$

$$\therefore \quad x_{A.C.} = 0.318 + 0.011 \doteqdot 0.33.$$

5. Rolling and Yawing Characteristics at low incidence

The Weissinger lifting line method has been used to obtain the additional span loading which results from an anti-symmetric distribution of incidence, and from such loadings the damping derivative in roll (ℓ_p) has been calculated in references 22 and 23. From curves given in these references the variation of ℓ_p with changes in planform parameters has been plotted in figure 11. On comparing these values of ℓ_p with the results of some unpublished calculations using the Falkner 21 vortex 3 point solution (see table 5 and figure 12) it is seen that good agreement exists between the results of the two methods.

In using the results of figure 11 it must be remembered that the section lift curve slope has been taken as 2π, and the following correction has been suggested[22,23]

$$\ell_p = (\ell_p)_{\text{fig.11}} \left[\frac{\text{Actual section lift curve slope}}{2\pi} \right]$$

Comparison with experimental results (figure 12 and table 6) gives support to this suggestion which is accordingly recommended.

Use has been made of unpublished calculations using Falkner's 21 vortex 3 point solution to predict variations with planform parameters of the damping derivative in yaw (n_{r_i}, due to induced drag) and of the yawing moment due to rolling (n_p^i). These variations are shown in figures 13 and 14. Quantitatively, the results, which are based on linear section lift data, will probably suffer from some inaccuracies. However, it is thought that the trends suggested are correct.

6. Conclusions

In general, fair agreement has been found to exist when the results of Weissinger's swept lifting line theory have been compared with those of other theories and with a limited number of experimental results, and, where possible, this comparison has been used to derive corrections to the results of Weissinger's theory. It is believed that the resulting diagrams provide a rapid method for estimating the particular characteristics of delta wings with an accuracy considered adequate in the design stage of an aeroplane.

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22.	J.D. Bird	Some theoretical low speed span loading characteristics of swept wings in roll and sideslip. N.A.C.A. T.N. 1839. Dec. 1948.
23.	J. De Young	Theoretical antisymmetric span loading for wings of arbitrary planform at subsonic speeds. N.A.C.A. T.N. 2140. July 1950.
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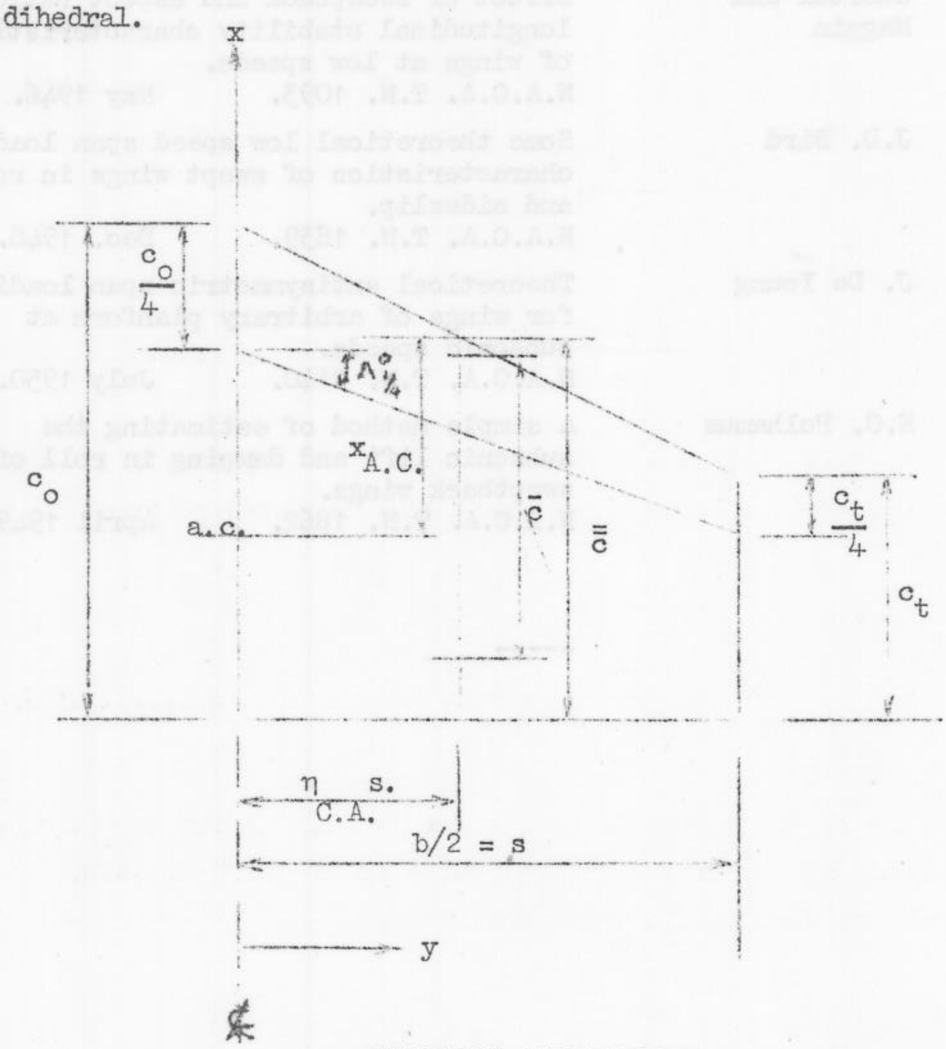
/APPENDIX ...

APPENDIX

Mean chord definitions and relationships

The following formulae apply to any straight tapered sweptback wing which is uncambered and untwisted, and has zero dihedral.



PLANFORM PARAMETERS

Let λ = taper ratio = $c_t/c_o = 1 - \tau$, say. The standard mean chord (S.M.C.) [or geometric mean chord (G.M.C.)] is defined as,

$$\bar{c} \equiv \frac{\text{gross wing area}}{\text{span}} \qquad \text{..........(1A)}$$

$$= \frac{S}{b}$$

$$= c_o/2 \; (1 + \lambda) \qquad \text{..........(2A)}$$

The centroid-of-area chord (C.A.C.) is defined as,

$$\bar{\bar{c}} = \frac{\int_{-1}^{1} c^2 \, d\eta}{\int_{-1}^{1} c \, d\eta} \qquad \text{..........(3A)}$$

/Since ...

Since the wing is assumed symmetrical in planform about the centre line,

$$\bar{\bar{c}} = \frac{\int_0^1 c^2 \, d\eta}{\int_0^1 c \, d\eta} = c_o \frac{\int_0^1 (1-\eta\tau)^2 d\eta}{\int_0^1 (1-\eta\tau) \, d\eta}$$

Hence

$$\bar{\bar{c}} = c_o \frac{(1-\tau+\frac{\tau^2}{3})}{(1-\frac{\tau}{2})}$$

$$= \frac{2}{3} \, c_o \, \frac{(1+\lambda+\lambda^2)}{(1+\lambda)} \qquad \text{..........(4A)}$$

A relationship between the lengths of the standard mean chord and centroid of area chord can easily be obtained from equations (2A) and (4A).

Thus,

$$\frac{\bar{c}}{\bar{\bar{c}}} = \frac{3(1+\lambda)^2}{4(1+\lambda+\lambda^2)} \qquad \text{..........(5A)}$$

ie $\bar{c} > \bar{\bar{c}}$

These two reference chords are located at the same spanwise station, namely at the centroid of area of the half wing. Their spanwise position is therefore given by

$$\eta_{C.A.} \cdot s = \frac{\int_0^s c\eta s \, dy}{\int_0^s c \, dy} = \frac{c_o \, s^2 \int_0^1 \eta(1-\eta\tau) d\eta}{c_o \, s \int_0^1 (1-\eta\tau) d\eta}$$

$$= \frac{s(\frac{1}{2}-\frac{\tau}{3})}{(1-\frac{\tau}{2})}$$

$$\eta_{C.A.} = \frac{1+2\lambda}{3(1+\lambda)} \qquad \text{..........(6A)}$$

/TABLE 1 ...

TABLE 1

LIFT CURVE SLOPE VALUES GIVEN BY LIFTING SURFACE THEORY

REF. NO.	REPORT NO. AUTHOR, ETC.	WING NO.	ASPECT RATIO A	TAPER RATIO λ	SWEEP-BACK $\Lambda^o_{\frac{1}{4}}$	THEORY DUE TO.-	$\left(\partial C_L/\partial \alpha\right)_{C_L = 0}$ THEORY	$\left(\partial C_L/\partial \alpha\right)_{C_L = 0}$ FROM FIG.1
-	UNPUBLISHED		1.0	0	71.57	Falkner	1.253	1.191
	CALCULATIONS		2.0	0	56.32	21 Vortex,	2.180	2.182
	'		3.0	0	45.00	3 Point Solution.	2.865	2.830
	'		4.0	0	36.87	'	3.360	3.339
	'		1.0	0.10	67.83	'	1.336	1.350
	'		2.0	0.10	50.87	'	2.320	2.286
	'		3.0	0.10	39.30	'	2.990	2.985
	'		4.0	0.10	31.53	'	3.515	3.500
	'		1.0	0.20	63.43	'	1.386	1.425
	'		2.0	0.20	45.00	'	2.370	2.360
	'		3.0	0.20	33.70	'	3.100	3.085
	'		4.0	0.20	26.57	'	3.660	3.595
	'		1.0	0.30	58.23	'	1.400	1.445
	'		2.0	0.30	38.92	'	2.420	2.402
	'		3.0	0.30	28.33	'	3.130	3.130
	'		4.0	0.30	22.00	'	3.630	3.640
3	ARC 12,222 H.C. Garner		3.0	0.143	37.0	Garner Method (c)	3.038	3.030
1	ARC 9,830 V.M.Falkner		2.31	0	52.4	Falkner, 21 V. 3 Pnt. Soln.	2.402	2.410
	'		2.31	0	52.4	Falkner 126V 6 Pnt. Soln. (Uncorr.)	2.518	2.410
9	KTH-Aero	A0	2.50	0	50.2	Falkner	2.62	2.54
	T.N. 10	B0	1.67	0	61.0	126 Vortex	1.97	1.90
	'	C0	1.00	0	71.6	6 Pnt. Soln.	1.31	1.191
	'	A2	1.34	0.303	50.2	(corrected	1.84	1.825
	'	B2	0.89	0.303	61.02	for disconty.	1.32	1.305
	'	C2	0.54	0.303	71.40	at centre line)	0.84	0.825
10	ARC 11,542		4.0	0	36.87	Falkner	3.47	3.339
	V.M.Falkner		3.0	0.138	36.87	126 Vortex, 6 Point.	3.14	3.025
	'		2.31	0.268	36.87	(uncorr.)	2.76	2.65

TABLE 2

EXPERIMENTAL VALUES OF LIFT CURVE SLOPE - WING ALONE

REF. NO.	REPORT NO. AUTHOR, ETC.	WING NO.	ASPECT RATIO A	TAPER RATIO λ	SWEEP-BACK $\Lambda^{o}_{\frac{1}{4}}$	R.N.x10^6 (Based on SMC)	AEROFOIL SECTION	$(\partial C_L/\partial \alpha)_{C_L = 0}$ EXPT.	FROM FIG.1
9 and 11	KTH - Aero	A0	2.50	0	50.2	1.02	FFA 104-5106	2.59	2.54
	T.N.4 and 10	B0	1.67	0	61.0	1.02	(symmetrical)	1.92	1.90
	S.B.Berndt and	C0	1.00	0	71.6	1.02	$(t/c)_{max}$=10%	1.31	1.191
	K.Orlik-Rückemann	A2	1.34	0.303	50.2	1.33	at 40% c	1.85	1.825
		B2	0.89	0.303	61.02	1.33	"	1.31	1.305
	"	C2	0.54	0.303	71.4	1.33	"	0.80	0.825
12	NACA TM 1176	FD	3.0	0	45.00	1.35	NACA 0012	2.807	2.830
	Lange/Wacke	ED	2.0	0	56.32	1.66	"	2.140	2.182
	"	DD	1.33	0	66.03	2.03	"	1.525	1.560
	"	CD	1.0	0	71.57	2.35	"	1.202	1.191
13	NACA TM 1225	DT½	1.33	0.50	36.87	2.03	NACA 0012	1.86	1.80
	Lange/Wacke	DT¼	1.33	0.25	53.47	2.03	"	1.83	1.825
	"	DT⅛	1.33	0.125	60.25	2.03		1.69	1.75
14	RAE Translation 276.H.Voepel	ET	2.0	0.33	37.2	~1.7	NACA 0012	2.50	2.407
15 and 16	RAE TN AERO.1869	1	4.0	0	36.87	2.1	Squire 'C'	3.44	3.339
	and Rep. 2284	2	3.0	0.138	36.87	2.4	$(t/c)_{max}$=10%	3.15	3.025
	Lock & others	3	2.31	0.268	36.87	2.7	at 35% c.	2.69	2.65
17	ARC 11,354	Δ1	3.87	0	37.8	3.6	Symmetrical	3.23	3.27
	Jones, Miles,	Δ1	3.04	0.143	36.6	4.0	(t/c) = 10%	3.03	3.06
	and Pusey	Δ1	2.38	0.268	36.0	2.4	at 35% c	2.65	2.698
	"	Δ2	2.31	0	52.4	4.1	"	2.39	2.41
18	NACA.TN.1468	11	3.0	0	45.0	0.3	Flat Plate	2.863	2.84
	L.P. Tosti	12	2.0	0	56.3	0.44	Thickness = ¾"	2.19	2.18
	"	13	1.0	0	71.6	0.62	Nose Rad. = ⅜"	1.36	1.191
	"	14	0.5	0	80.4	0.87	T.E.Thick. = 1/16"	0.62	0.61
	"	15	2.0	0.20	45.0	0.41	T.E.Angle = 9.8°	2.67	2.36
	"	16	1.0	0.50	45.0	0.53		1.55	1.43

TABLE 3

EXPERIMENTAL RESULTS FOR STALLING AND

PITCHING-MOMENT CURVES AT THE STALL (WING ALONE)

REF. NO.	REPORT NO. AUTHOR, ETC.	WING NO.	ASPECT RATIO A	TAPER RATIO λ	SWEEP-BACK $\Lambda^o_{\frac{1}{4}}$	$RN \times 10^{-6}$ (Based on SMC)	AEROFOIL SECTION	$C_{L_{max}}$	$\alpha^o_{C_{L_{max}}}$	C_M RATING AT STALL	SYMBOL
12	NACA	FD	3.0	0	45.0	1.35	NACA 0012	0.913	25.6	Stable	▽
	TM. 1176	ED	2.0	0	56.32	1.66	'	1.165	38.0	Marginal	⊙
	Lange /	DD	1.33	0	66.03	2.03	'	1.030	39.0	'	⊙
	Wacke	CD	1.0	0	71.57	2.35	'	0.940	38.0+	'	⊙
13	NACA TM.1225	DT $\frac{1}{8}$	1.33	.125	60.25	2.03	NACA 0012	1.20	30.8	Marginal	⊙
	Lange	DT$\frac{1}{4}$	1.33	.250	53.47	2.03	'	1.21	36.0	Stable	▽
	/ Wacke	DT$\frac{1}{2}$	1.33	.500	36.87	2.03	'	1.15	38.0	Stable	▽
14	RAE Library Translation 276	ET	2.0	.33	37.2	1.7	NACA 0012	0.862	21.0	Stable	▽
15	RAE. T.N.	1	4.0	0	36.87	2.1	Squire 'C'	0.868	20.6	Stable	▽
	Aero. 1869	2	3.0	.138	36.87	2.4	$(t/c)_{max}$=10%	0.88	20.6	'	▽
	Lock & Others	3	2.31	.268	36.87	2.7	at 35% c	0.86	20.6	'	▽
18	NACA	11	3.0	0	45.0	0.3	Flat Plate	0.98	25.5	Stable	▽
	TN. 1468	12	2.0	0	56.3	0.44	Thickness	1.267	34.5	Stable	▽
	L.P.Tosti	13	1.0	0	71.6	0.62	= 3/4in.	1.04	34.0	Unstable	▲
		14	0.5	0	80.4	0.87	Nose Rad-	0.87	34.0	Unstable	▲
		15	2.0	.20	45.0	0.41	ius = $\frac{3}{8}$"	1.00	24.0	Stable	▽
		16	1.0	.50	45.0	0.53	TE.Thick-ness=1/16"	1.35	32.0	Stable	▽
		17	0.33	.50	71.6	0.93		0.99	36.0	Marginal	⊙
		18	0.176	.50	80.4	1.31	TE.Angle =9.8°	0.78	38.0	Unstable	▲
17	ARC 11354	Δ1	3.87	0	37.8	3.6	Symmetri-cal	0.895	20.0	Stable	▽
	Jones, Miles,	Δ1	3.04	.143	36.6	4.0	$(t/c)_{max}$=10%	0.88	20.4	'	▽
	and Pusey	Δ1	2.38	.268	36.0	2.4	at 35%c	0.925	19.8	'	▽
		Δ2	2.31	0	52.4	4.1		1.13	32.4	'	▽
19	NACA TN.1650 B.H. Wick		2.0	0	56.3	1.8	NACA 0012	1.24	39.0	*	

* Result from pressure distribution - experimental

\+ $C_{L_{max}}$ constant with α after this value.

TABLE 4

EXPERIMENTAL VALUES OF THE AERODYNAMIC CENTRE LOCATION ($x_{A.C.}$), AND DATA FOR ERROR CURVES

REF. NO.	REPORT NO. AUTHOR, ETC.	WING NO.	ASPECT RATIO A	TAPER RATIO λ	SWEEP-BACK $\Lambda_{\frac{1}{4}}^{o}$	RNx10^{-6} (based on SMC)	AEROFOIL SECTION	$x_{A.C.}$ (Expl.)	$x'_{A.C.}$ * (From Fig.7)	$\Delta x'_{A.C.}$ as % of *
15	RAE. TN. Aero 1869	1	4.0	0	36.87	2.1	Squire 'C'	.340	.329	3.34
and	& Rep. 2284	2	3.0	.138	36.87	2.4	$(t/c)_{max} = 10\%$	.305	.292	4.45
16	Lock & Others	3	2.31	.268	36.87	2.7	at 35% c.	.286	.267	7.12
12	NACA TM. 1176	FD	3.0	0	45.00	1.35	NACA 0012	.344	.336	2.38
		ED	2.0	0	56.32	1.66	'	.374	.342	9.35
	Lange / Wacke	DD	1.33	0	66.03	2.03	'	.393	-	-
		CD	1.0	0	71.57	2.35	'	.411	-	-
11 and 9	KTH-Aero TN. 4 & 10	AO	2.50	0	50.2	1.02	FFA 104-5106	.363	.339	4.72
		BO	1.67	0	61.0	1.02	$(t/c)_{max} = 10\%$ at 40% c	.383	.343	11.65
18	NACA TN. 1468	11	3.0	0	45.0	0.3	Flat	.340	.336	1.2
		12	2.0	0	56.3	0.44	Plate	.350	.342	2.3
	L.P. Tosti	15	2.0	.20	45.0	0.41	'	.306	.281	8.9
19	NACA TN. 1650 B.H. Wick		2.0	0	56.3	1.8	NACA 0012	.390	.342	14.6
17	ARC. 11,354	Δ1	3.87	0	37.8	3.6	Symmetrical	.331	.331	0
	Jones, Miles,	Δ1	3.04	.143	36.6	4.0	$(t/c)_{max} = 10\%$	.303	.292	3.8
	& Pusey	Δ1	2.38	.268	36.0	2.4	at 35% c	.271	.267	1.5
		Δ2	2.31	0	52.4	4.1		.367	.341	7.6

TABLE 5

DAMPING-IN-ROLL DERIVATIVE $(-\ell_p)$.

VALUES GIVEN BY LIFTING SURFACE THEORY

REPORT NO. AUTHOR, ETC.	ASPECT RATIO A	TAPER RATIO λ	SWEEPBACK $\Lambda^{o}_{\frac{1}{4}}$	THEORY DUE TO	$(-\ell_p)$ THEORY	$(-\ell_p)$ FROM FIG. 11
Unpublished	1.0	0	71.57	Falkner;	.0842	.080
Calculations	2.0	0	56.32	21 Vortex,	.1505	.150
'	3.0	0	45.00	3 Point	.2037	.205
'	4.0	0	36.87	Solution.	.2468	.247
'	1.0	0.1	67.83	'	.0904	.090
'	2.0	0.1	50.87	'	.1690	.165
'	3.0	0.1	39.30	'	.2287	.228
'	4.0	0.1	31.53	'	.2758	.280
'	1.0	0.2	63.43	'	.0944	.095
'	2.0	0.2	45.00	'	.1761	.175
'	3.0	0.2	33.70	'	.2422	.241
'	4.0	0.2	26.57	'	.2943	.296
'	1.0	0.3	58.23	'	.0955	.097
'	2.0	0.3	38.92	'	.1799	.180
'	3.0	0.3	28.33	'	.2499	.250
'	4.0	0.3	22.00	'	.3065	.308

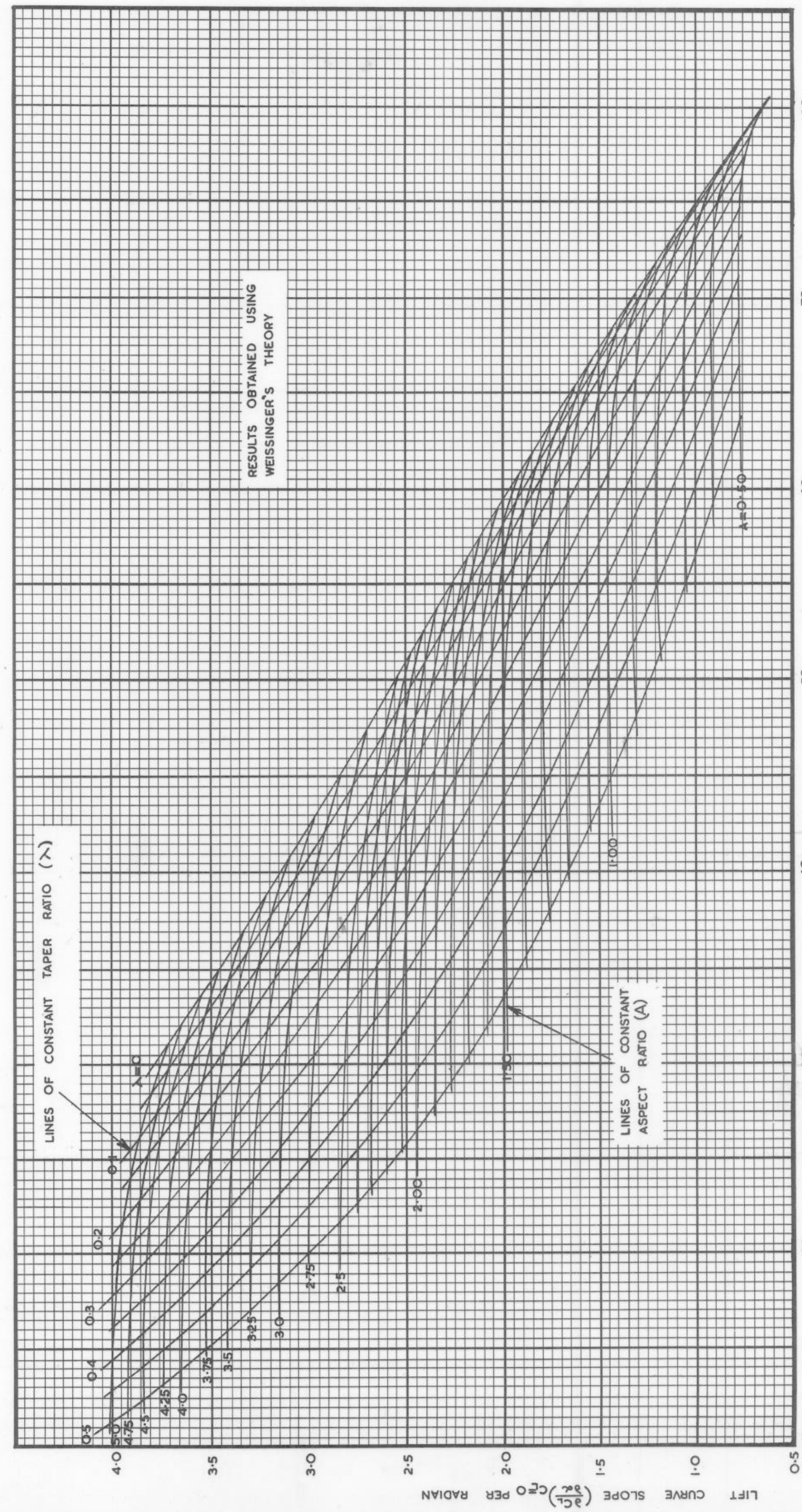
TABLE 6

EXPERIMENTAL VALUES OF DAMPING-IN-ROLL

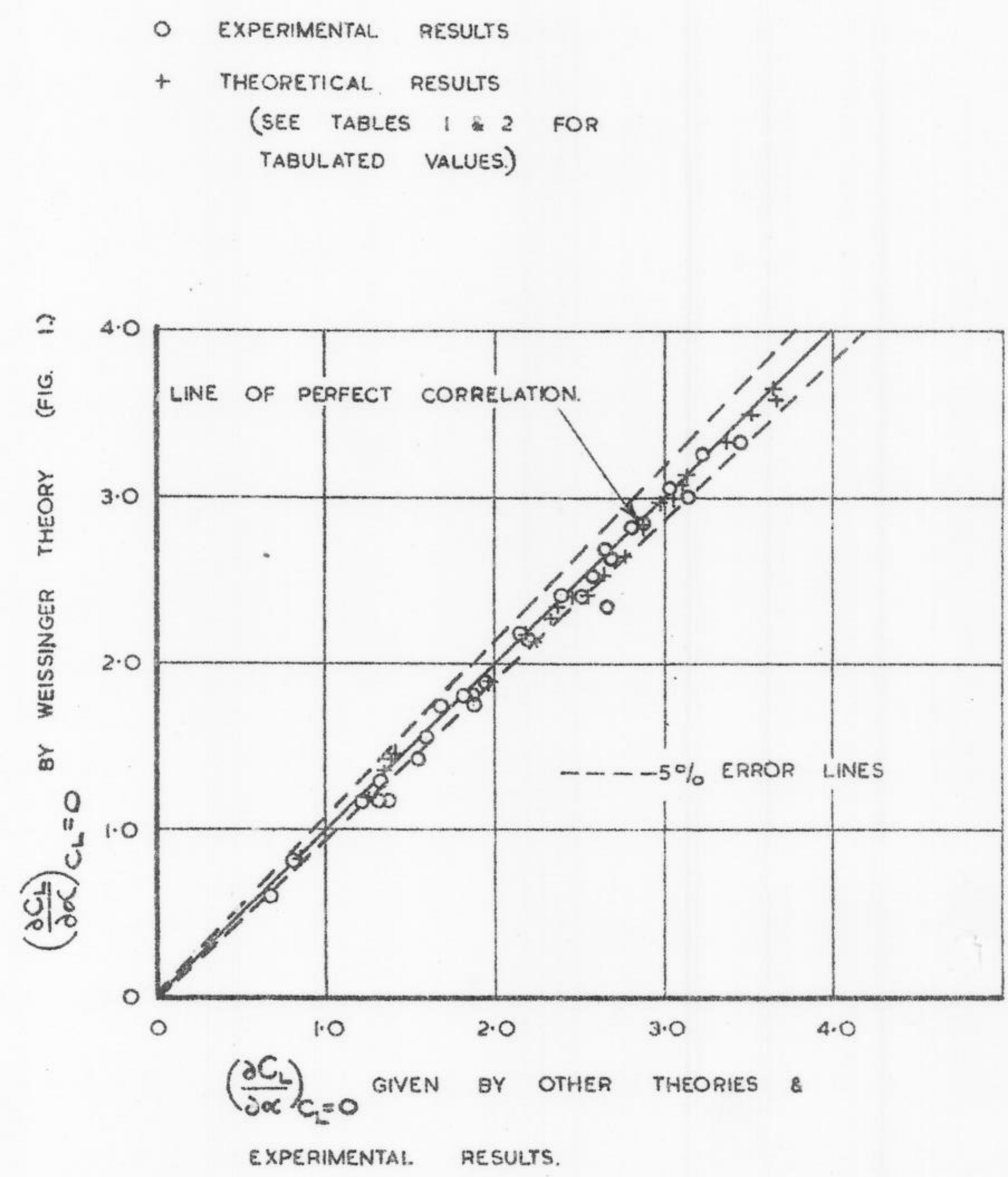
DERIVATIVE $(-\ell_p)$ (WING ALONE)

REF. NO.	REPORT NO. AUTHOR, ETC.	WING NO.	ASPECT RATIO A	TAPER RATIO λ	SWEEP-BACK $\Lambda^{o}_{\frac{1}{4}}$	$RN \times 10^{-6}$ (Based on SMC)	AEROFOIL SECTION	$(-\ell_p)$ EXP.	$(-\ell_p)$ FROM FIG. 8
18	NACA TN. 1468	11	3.0	0	45.0	0.3	Flat Plate	.166	.205
	L.P.Tosti	12	2.0	0	56.3	0.44	Thickness $=\frac{3}{4}''$	.112	.150
		13	1.0	0	71.6	0.62	Nose Rad. $=\frac{3}{8}''$	.036	.080
		14	0.5	0	80.4	0.87	TE. Thick. $=\frac{1}{16}''$	.012	.040
		15	2.0	0.20	45.0	0.41	TE. Angle=9.8°	.140	.174
		16	1.0	0.50	45.0	0.53		.105	.100
24	NACA TN. 1862	5	4.0	0	36.87		NACA 0012	.228	.247
	E.C.Polhamus	6	3.0	0.15	36.5		(approx.)	.229	.235
		15	2.31	0	52.4		NACA 0012	.150	.170
		19	1.07	0	70.0			.075	.088

FIG. I.

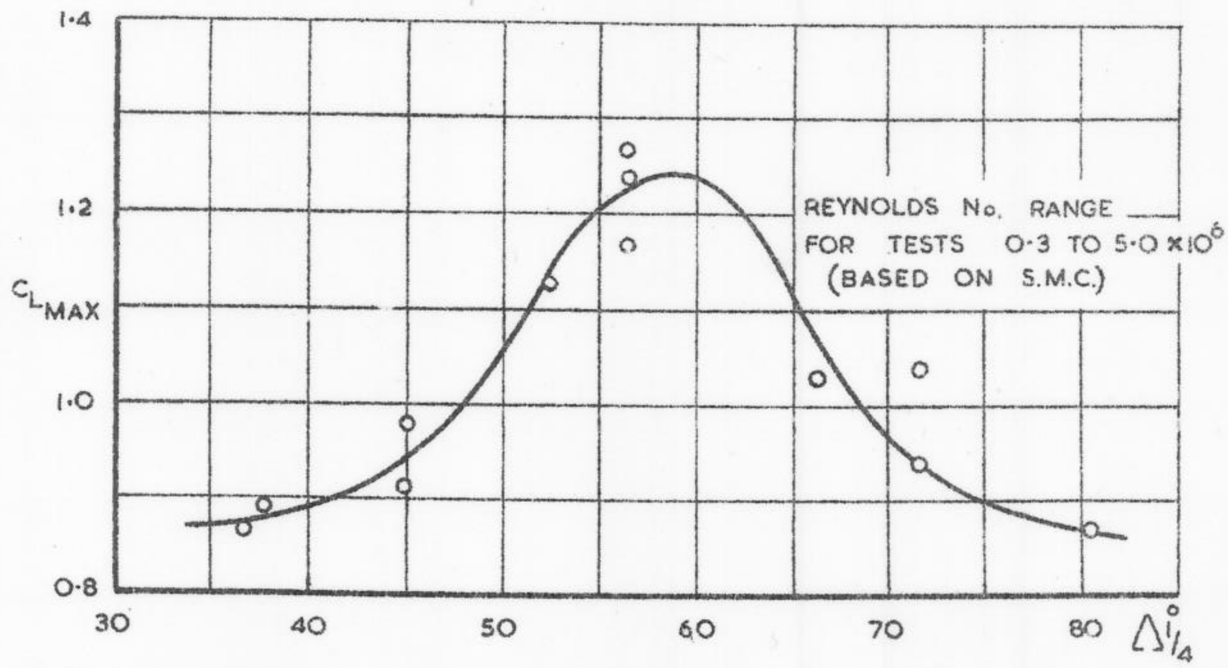


VARIATION OF LIFT CURVE SLOPE WITH PLANFORM PARAMETERS, FOR DELTA WINGS.



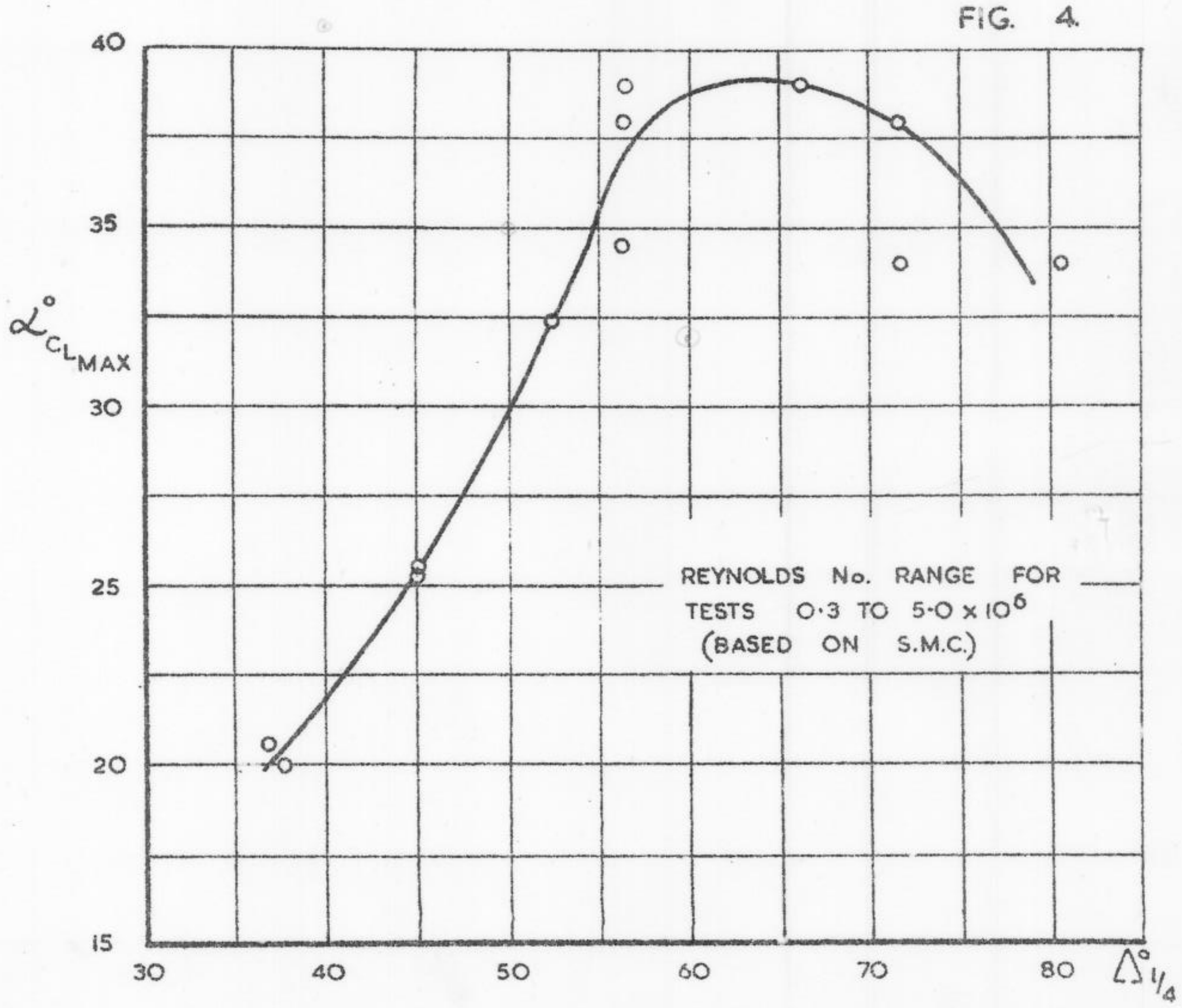
CORRELATION OF LIFT CURVE SLOPE BY WEISSINGER THEORY WITH VALUES GIVEN BY OTHER THEORIES & EXPERIMENT

FIG. 3.



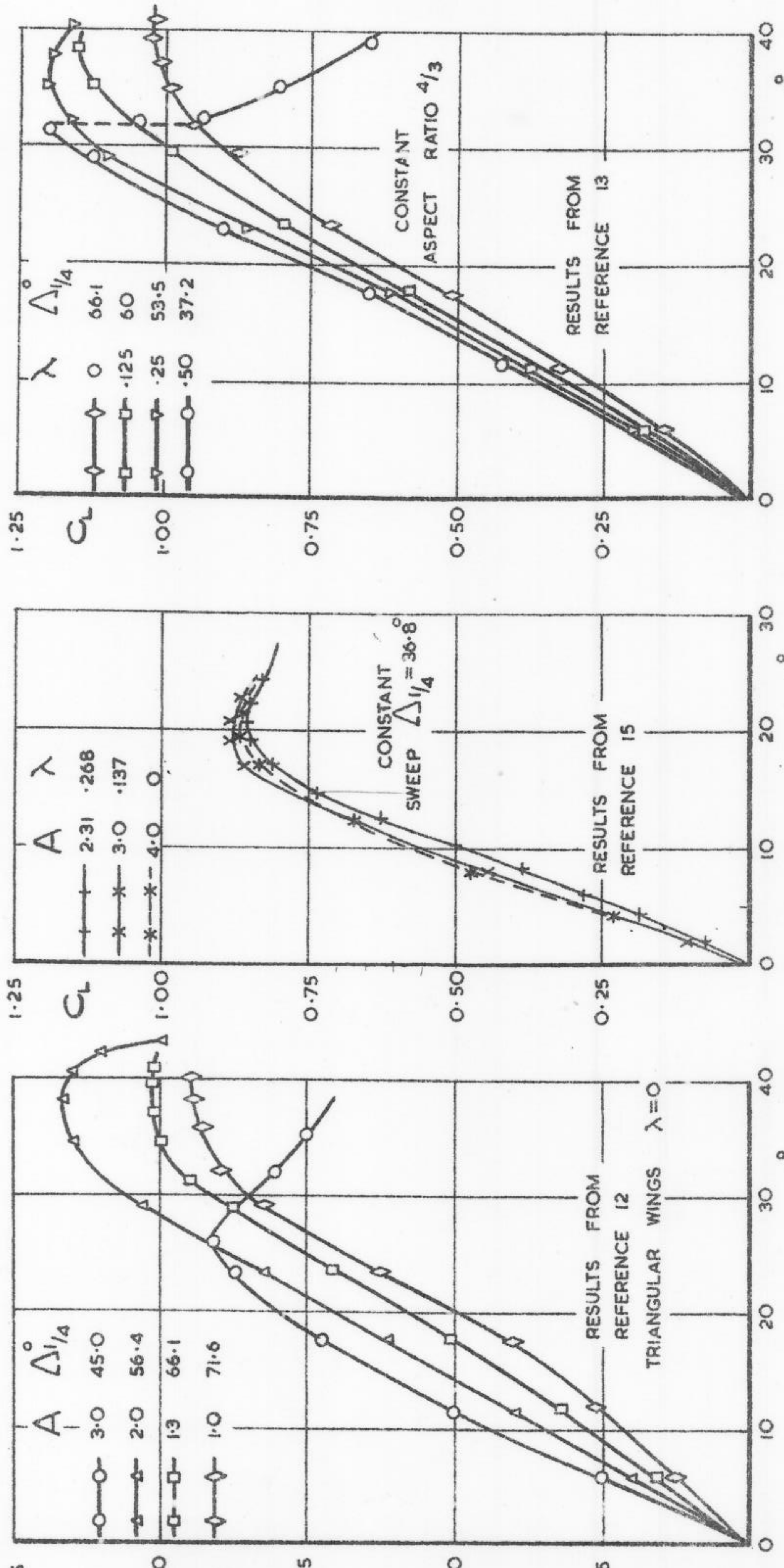
VARIATION OF MAX. LIFT COEFFICIENT WITH SWEEPBACK (AND ASPECT RATIO) FOR DELTA WINGS OF TAPER RATIO = 0 (TRIANGULAR WINGS.)

FIG. 4.



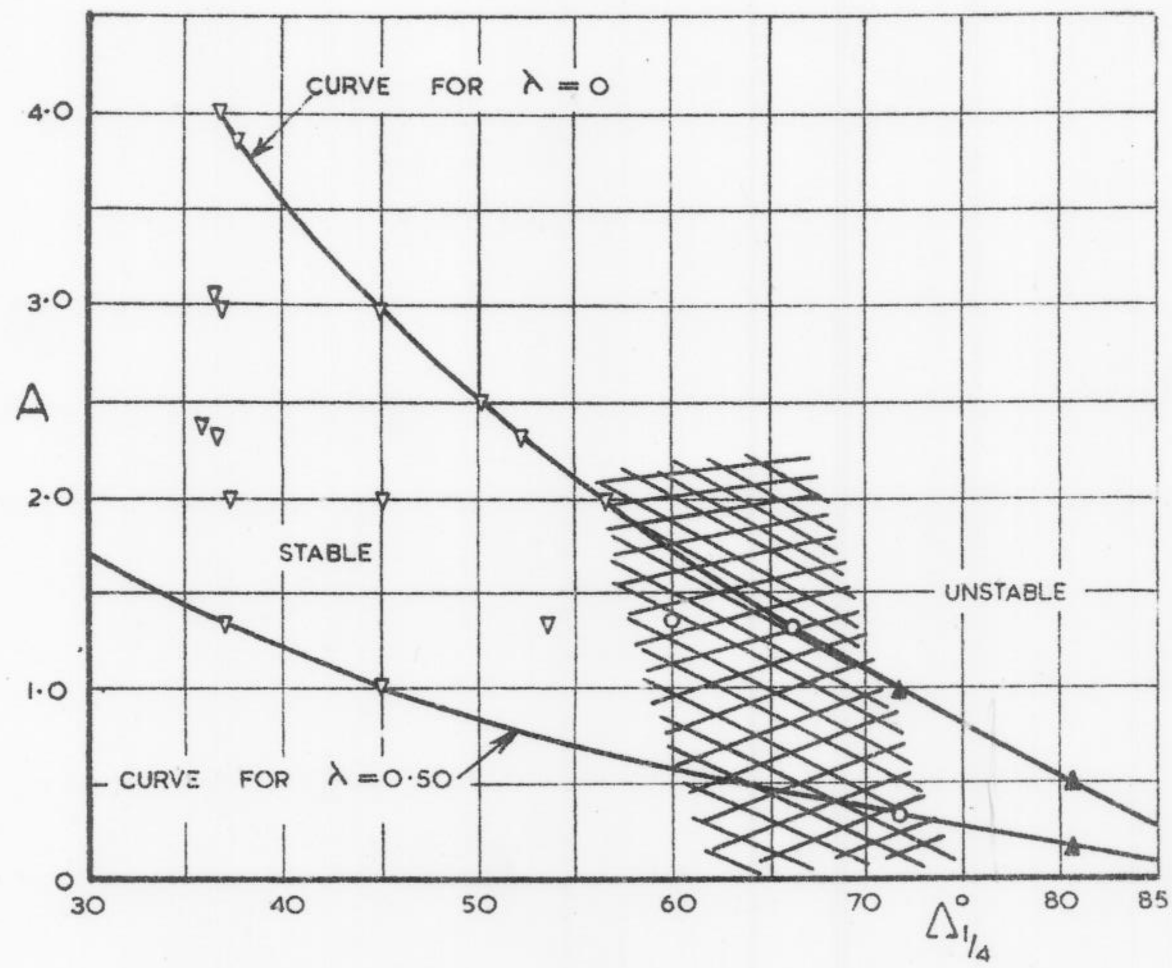
VARIATION OF INCIDENCE AT MAX. LIFT COEFF. WITH SWEEPBACK (AND ASPECT RATIO) FOR DELTA WINGS OF TAPER RATIO = 0 (TRIANGULAR WINGS)

FIG. 5.

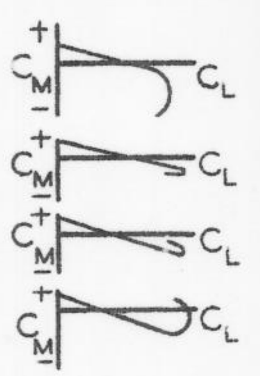


TYPICAL LIFT CURVES FOR DELTA WINGS.

FIG. 6.



TYPE OF C_M-C_L CURVE AT STALL — RATING — SYMBOL



STABLE ▽

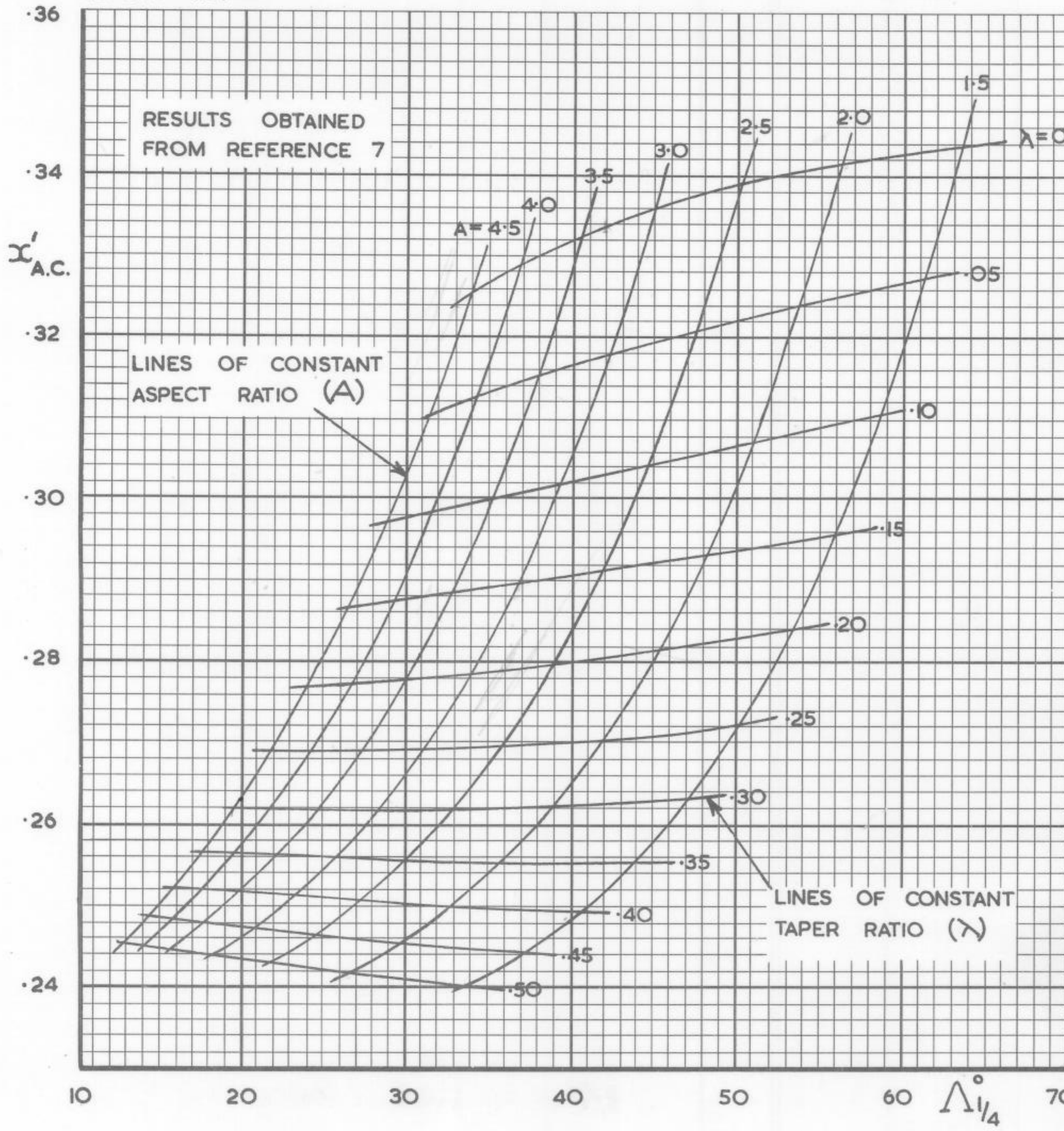
MARGINAL ○

UNSTABLE ▲

SEE TABLE 3 FOR ORIGIN OF RESULTS.

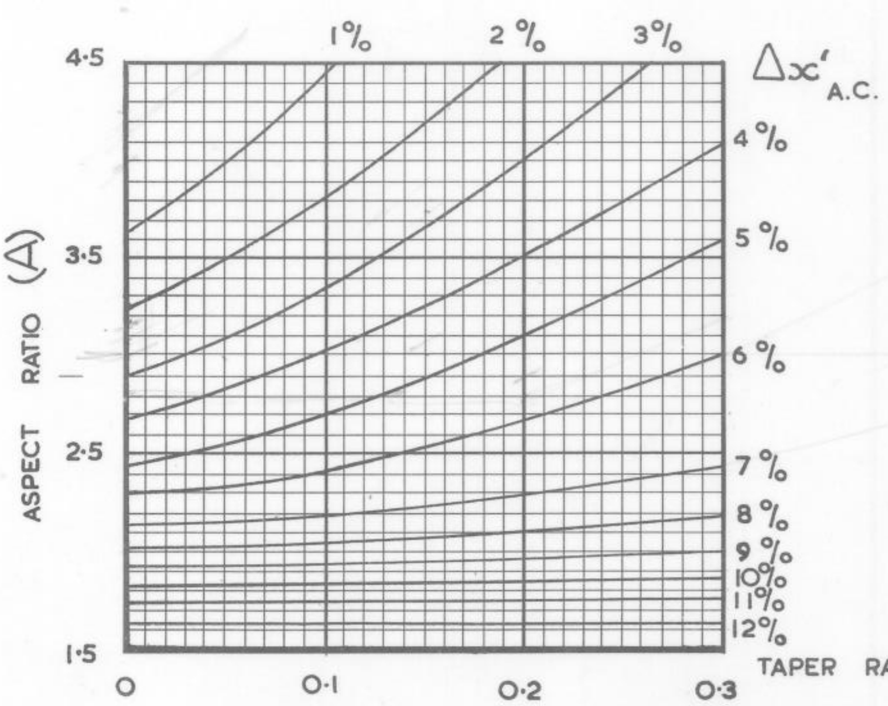
EFFECT OF PLANFORM PARAMETERS ON LONGITUDINAL STABILITY CHARACTERISTICS OF DELTA WINGS NEAR THE STALL

FIG. 7.



FIRST APPROXIMATION OF POSITION OF AERODYNAMIC CENTRE.

FIG. 8.



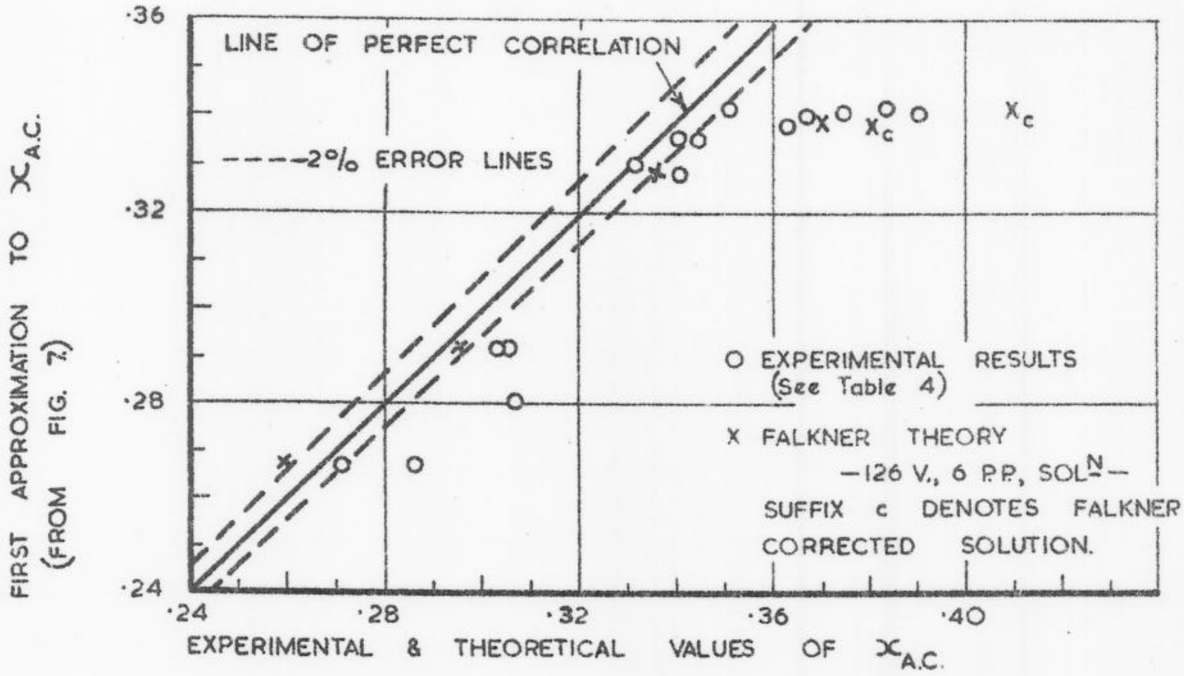
$\Delta x'_{A.C.}$ IS GIVEN AS A PERCENTAGE OF $x'_{A.C.}$

$$x_{A.C.} = x'_{A.C.} + \Delta x'_{A.C.}$$

SEE EXAMPLE IN PARAGRAPH 4.

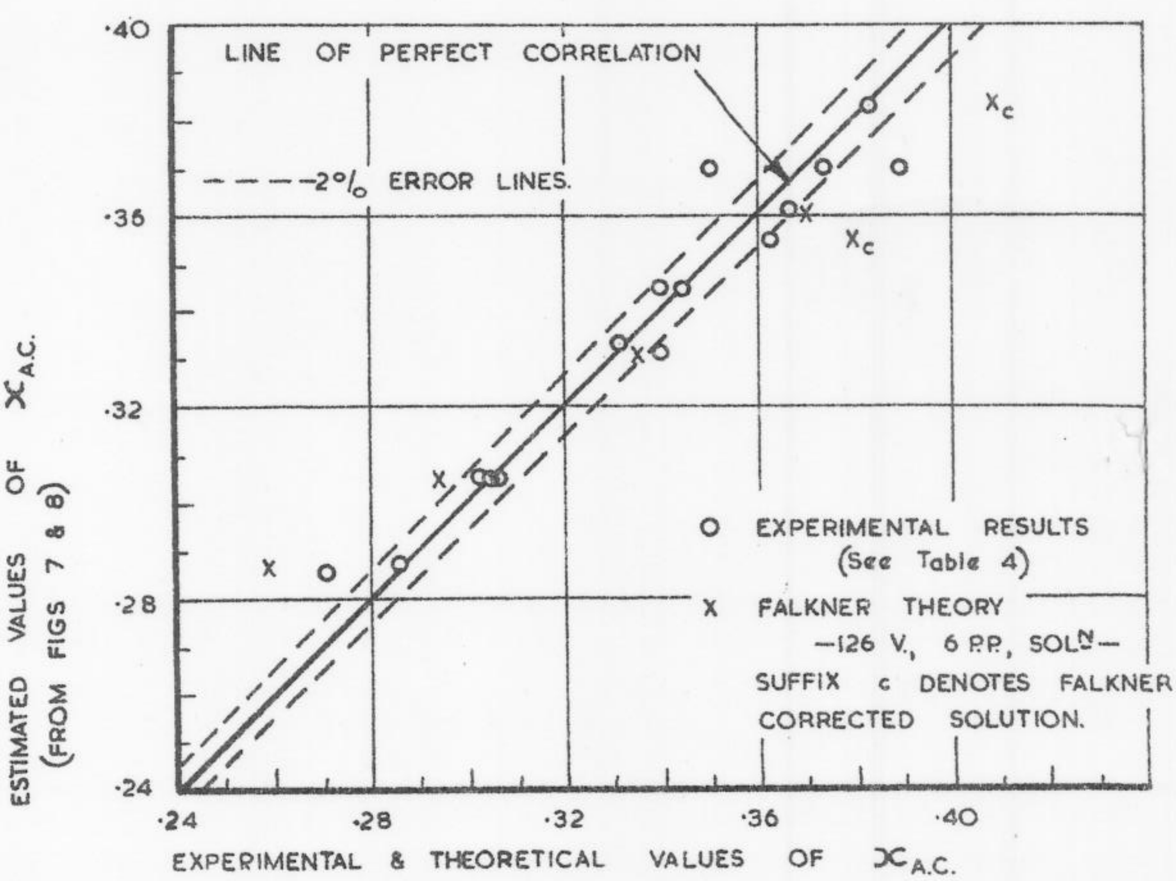
EMPIRICAL CORRECTION CURVES FOR LOCATION OF AERODYNAMIC CENTRE.

FIG. 9.



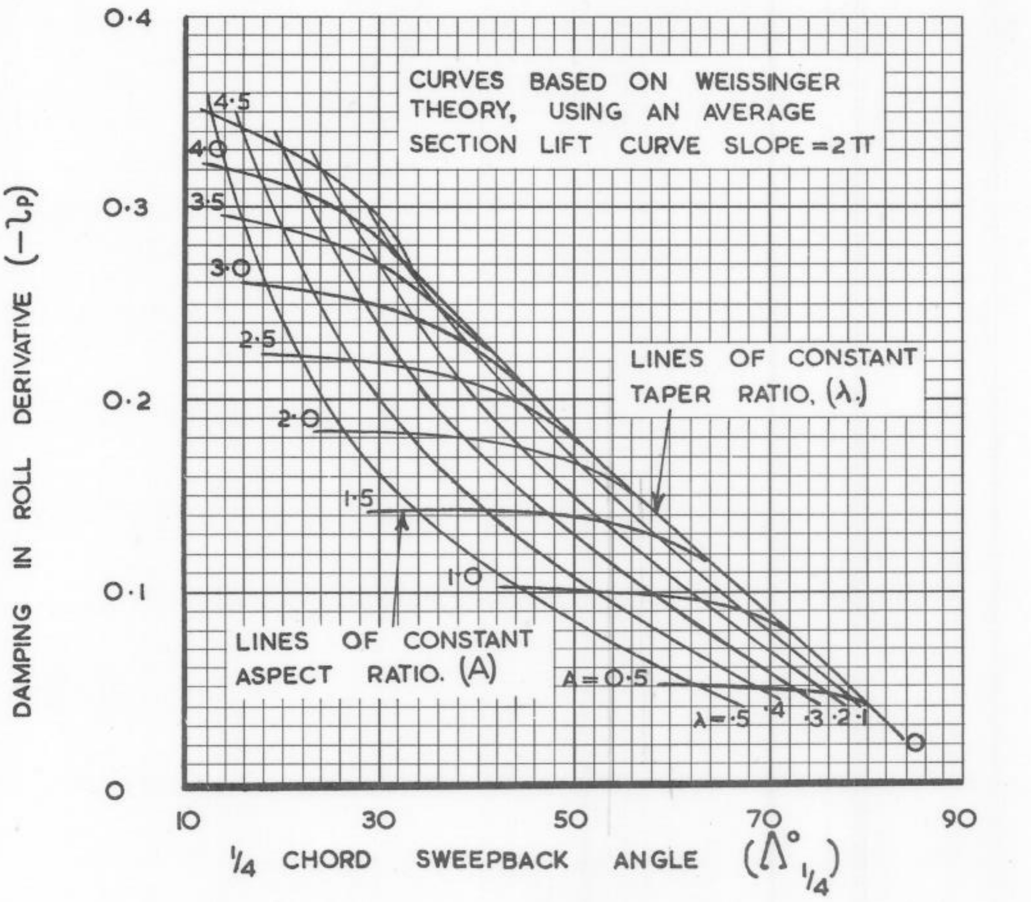
COMPARISON OF ESTIMATED $x_{A.C.}$ VALUES FROM FIRST APPROXIMATION (FIG. 7) WITH VALUES GIVEN BY FALKNER THEORY & EXPERIMENTAL RESULTS.

FIG. 10.



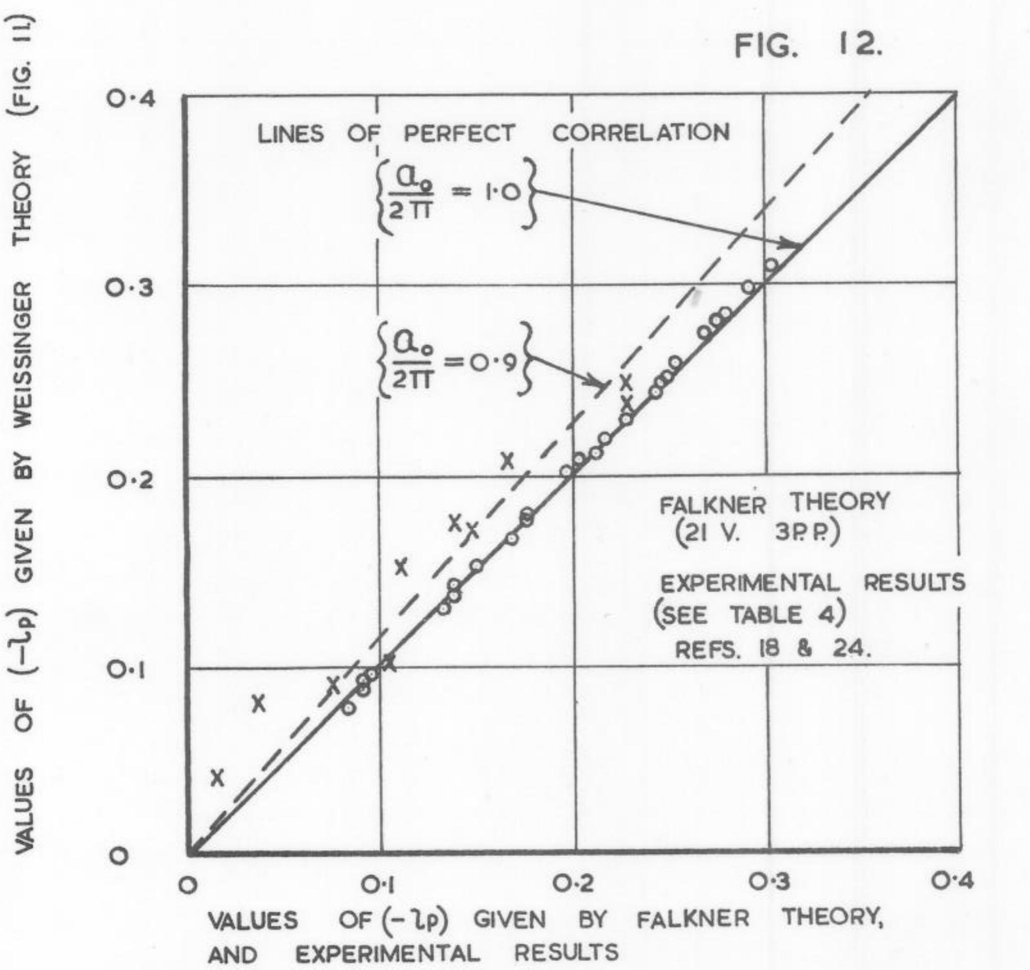
COMPARISON OF ESTIMATED A.C. LOCATION VALUES FROM FIGS. 7 & 8 WITH VALUES GIVEN BY FALKNER THEORY & EXPERIMENTAL RESULTS.

FIG 11.

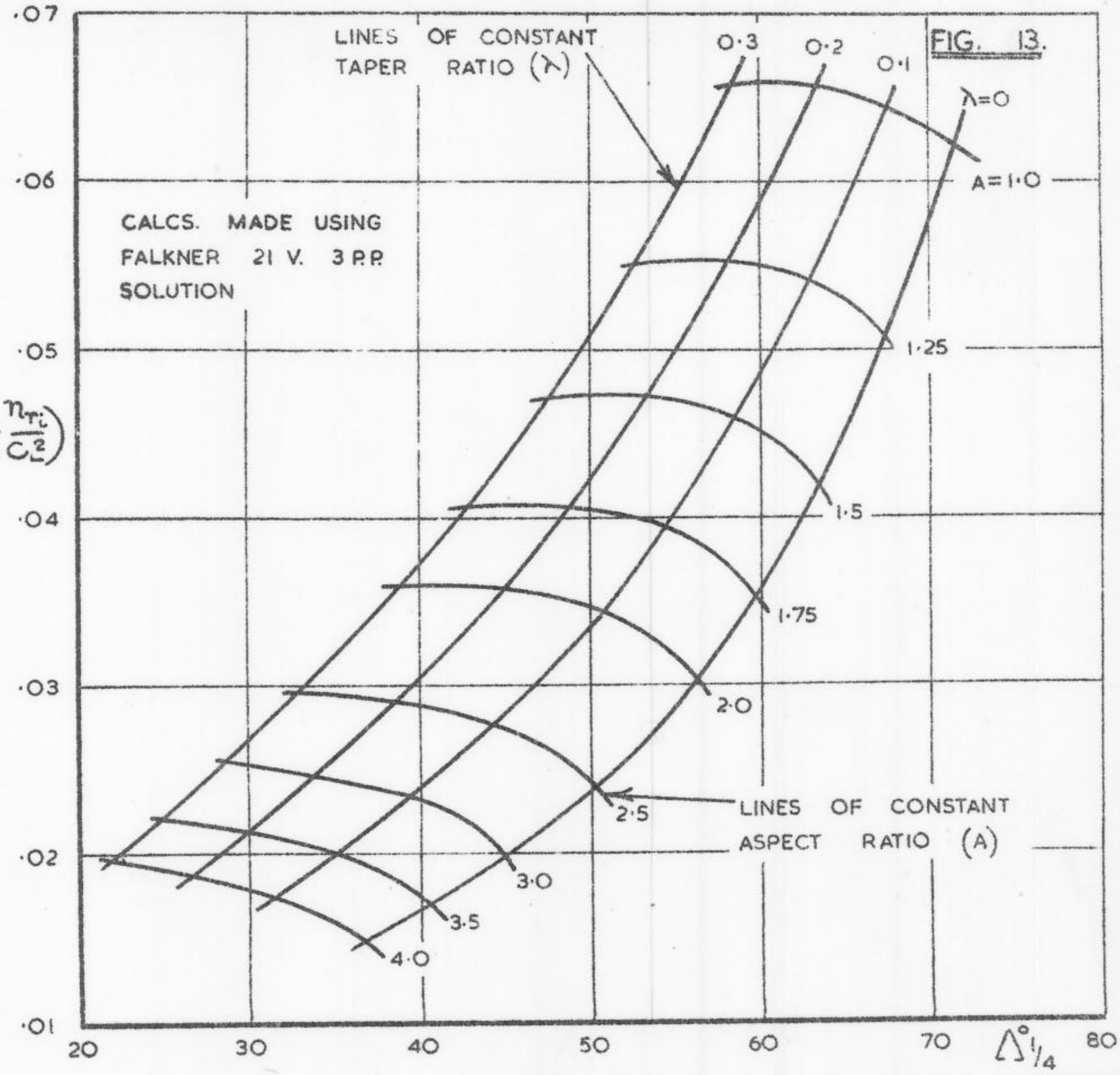


VARIATION OF DAMPING-IN-ROLL DERIVATIVE (l_p) WITH WING PLANFORM PARAMETERS FOR DELTA WINGS.

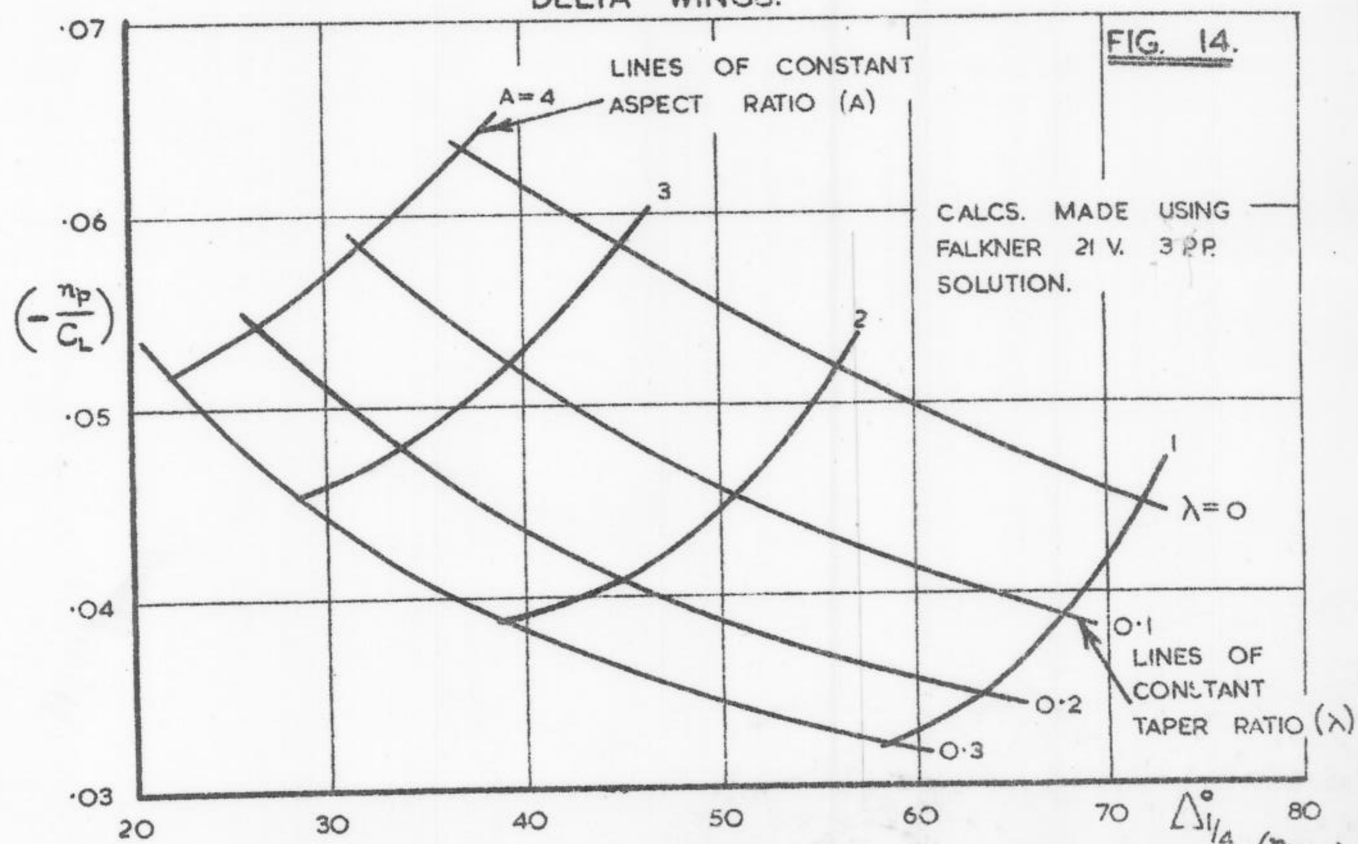
FIG. 12.



COMPARISON OF RESULTS WITH OTHER THEORIES AND EXPERIMENTS.



VARIATION OF YAWING MOMENT COEFF. DUE TO YAWING-INDUCED COMP$^{T.}$ $-\left(n_{r_i}/C_L^2\right)$ — WITH WING PLANFORM PARAMETERS FOR DELTA WINGS.



VARIATION OF YAWING MOMENT COEFF. DUE TO ROLLING $\left(n_p/C_L\right)$ WITH WING PLANFORM PARAMETERS FOR DELTA WINGS.